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RESEARCH MEMORANDUM

PERFORMANCE OF SUPERSONIC SCOOP INLETS

By M. I. Weinstein

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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RESEARCH MEMORANDUM

PERFORMANCE OF SUPERSONIC SCOOP INLETS

By M. I. Weinstein

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The design of efficient scoop inlets for turbojet-powered supersonic aircraft is complicated by the necessity for removing the boundary layer ahead of the inlets and for varying the geometry of the inlet to match the engine air induction requirements efficiently over a wide range of flight conditions. The design considerations for successfully coping with these problems have been broadly defined in recent reports by a number of investigators (references 1 to 6).

It is the purpose of this paper to discuss the effect of specific inlet design details on the performance of a series of basic scoop configurations. The data presented herein represent only a brief summary of the results of an extensive program conducted in the 8- by 6-foot supersonic tunnel at the Lewis laboratory. The investigation was conducted on a $\frac{1}{4}$-scale model of the forebody of the Douglas X-3 airplane shown in figure 1. Twin scoop inlets were located on either side of the fuselage aft of the pilot's canopy. Except where noted, the inlets were pointed downward slightly so that they were effectively alined with the entering flow at the airplane design cruise attitude of 3° angle of attack. The major portion of the approaching fuselage boundary layer was diverted around the inlets, passing outwards through the open sides of the ram-type boundary-layer bleed scoops. The remainder of the boundary layer was carried internally through constant-area ducts and exited at the rear of the model. As a consequence of fuselage shape, the Mach number ahead of the inlets at cruise angle of attack averaged about 0.2 below the free-stream value with negligible loss in free-stream total pressure. The Reynolds number based on model length ahead of the inlets was approximately 29×10^6 for supersonic speeds and 19×10^6 for subsonic speeds.

Investigations at the NACA Laboratories established (references 1 and 2) that, if the boundary layer is prevented from entering scoop inlets (specifically, half-conical-spike types), pressure recoveries can be realized which approach those for similar configurations used as

nose inlets. However, the data were not obtained from actual fuselage configurations and the drag penalty associated with removal of the boundary layer was not determined. From the data of the present investigation, it was possible to evaluate these drag costs using a realistic fuselage and, in addition, for a number of scoop inlet types. Typical data are shown in figure 2 for the case of a scoop inlet having a rounded lip on a semicircular cowl used in conjunction with a two-dimensional compression ramp. At the design Mach number of 2.0 and at 3° angle of attack, the optimum performance is here shown as a function of h/δ, where h is the height of the boundary-layer bleed scoop (and of the 2509
entering streamline) and δ is the boundary-layer thickness based on 99-percent velocity ratio. The pressure recovery P_2/P_0 is the ratio of total pressure at the diffuser exit (the engine compressor face) to free-stream total pressure. The drag coefficient C_D is based on the maximum frontal area of the configuration and represents the internal thrust minus the sum of the balance reading and the base force. Here the internal thrust is defined as the change in momentum of the flow entering the inlet measured between free-stream conditions and those at the diffuser exit. The over-all effect of boundary-layer removal is shown in the lowest curve as the ordinate $\Delta(F_n-D)/F_{n,ideal}$ which represents the improvement in engine thrust minus configuration drag obtained by increasing h/δ and is expressed as a fraction of the ideal engine thrust (or thrust at 100-percent pressure recovery). For calculation of thrust, a representative turbojet engine installation was assumed to be operating at an altitude of 35,000 feet and to include fully expanding nozzles on afterburners having 3900° R outlet temperatures. The drag penalty incurred by complete removal of the boundary layer is so small compared to improvement in pressure recovery, and hence thrust, that the net propulsive force is increased 16 percent of the ideal thrust. Subsequent data in this paper at 3° angle of attack will represent complete boundary-layer removal $\left(\frac{h}{\delta} = 1.0\right)$ except where noted.

For inlets intended to operate over a wide Mach number range, a design feature of considerable importance is the relative sharpness of the lip profile. Some experimental data concerning this problem are presented in reference 6. At the Ames Laboratory an experimental investigation was made (reference 7) to determine an efficient subsonic lip for this airplane. This lip design was employed in the initial phase of this investigation on an inlet designed for high subsonic performance. In figure 3 the performance of this rounded-lip inlet is compared with that of a sharp-lip inlet designed for high supersonic performance, both operating at design Mach number of 2.0 and at cruise angle of attack of 3°. Details of the inlets are shown in the figures at the right of the data. Both inlets had an approximately semicircular cowl and achieved variable geometry by changes in the angle of the compression ramp, in

this case fixed at 14° relative to the canopy surface. The internal and external lip angles of the sharp-lip inlet were designed to avoid shock detachment behind the oblique shock at this design free-stream Mach number. At the left side of the figure the performances are plotted as a function of the diffuser discharge Mach number M_2. The mass-flow ratio m_2/m_c represents the mass flow at the diffuser exit divided by the mass flow over the canopy that would pass through an area equal to the combined projected area of the inlet face and the compression surface. Thus, a
2509 mass ratio of unity represents the maximum mass flow that could be captured by an inlet. Pressure recovery and drag coefficient are as previously defined.

Both the sharp-lip inlet and the rounded-lip inlet gave essentially the same recoveries. The difference in drags between the inlets can be primarily attributed to the differences in additive drag. The inherent internal contraction of the rounded-lip inlet prevented the swallowing of the normal shock at the design Mach number of 2.0 which caused the resultant 35-percent supercritical mass-flow spillage and attendant higher drag. Although the sharp-lip inlet is spilling approximately 16 percent of the mass flow in the supercritical region, this spillage represents the inherent inability of a semicircular cowl to capture all of the flow compressed by the ramp. Reductions in this supercritical spillage can be accomplished by utilizing a rectangular cowl opening in conjunction with the two-dimensional compression ramp.

At the right of figure 4 are shown two inlets utilized in this investigation which had rectangular cowls and sharp-lip profiles. The corners of the cowls were slightly rounded. The upper edges of the cowl lips were positioned slightly behind the oblique shock emanating at Mach number 2.0 from the 12° compression ramp. On the inlet shown above, side fairings swept at this same shock angle were added in order to preserve the two-dimensionality of the flow compressed by the ramp. Performance of the inlets is again plotted as a function of diffuser discharge Mach number at design conditions.

The pressure recoveries of both rectangular cowl inlets were generally higher than the recoveries previously shown for the semicircular cowl with the sharp-lip profile. Side fairings generally improved the pressure-recovery characteristics of the rectangular cowl inlets but resulted in appreciable shock oscillations or pulsing in the subcritical region. The rectangular cowl inlets indicated considerable reduction in spillage compared to the sharp-lip, semicircular cowl inlet as can be seen by reference to the dashes showing the supercritical mass-flow ratio of the latter inlet. Side fairings reduced the spillage essentially to zero. Despite the relatively large difference in spillage, appreciable difference in drag between rectangular and semicircular inlets is not evident. This results from the fact that, behind oblique shocks, relatively little additive drag is associated with spillages of the order shown here.

The conical-spike type inlet, commonly used as a supersonic nose inlet, was also included in the investigation; its performance is shown in figure 5. This scoop configuration is the three-dimensional counterpart of the previously shown rectangular cowl inlet with side fairings since both types can be designed to spill zero mass flow and hence have zero additive drag. Variable geometry is achieved with this inlet by varying the longitudinal position of the spike with respect to the cowl. A sharp-lip profile was incorporated and was designed to avoid shock detachment in the conical flow field at the free-stream Mach number 2.0. 2509

Pressure recoveries of the conical spike inlet were generally lower than those of the rectangular cowl inlet with side fairings. Similar pronounced pulsing occurred in the subcritical regime. Of interest is the fact that the inlets other than the two shown here did not exhibit pulsing characteristics except at very low mass-flow ratios. Little difference is noticeable in the supercritical mass-flow spillages of the two inlets compared here. The slight drag reduction of the spike inlet is largely attributable to a decrease in pressure drag.

Neither pressure recovery nor drag alone can provide the proper criterion for comparative evaluation of the various inlets. Comparison can be made, however, on the basis of thrust minus drag F_n-D calculated from the indicated pressure recoveries and drag coefficients. Here again thrusts are derived from the previously considered turbojet engine installation operating at an altitude of 35,000 feet. The peak value of this parameter occurred for all inlets at an M_2 slightly less than critical as is indicated by a typical example in figure 5. The maximum value of F_n-D indicates the optimum M_2 for matching a given inlet to an engine and in addition provides the desired figure of inlet merit.

Relative performance of the various inlets at design $M_0 = 2.0$ and at $\alpha = 3^\circ$ is shown in figure 6. Comparison is made on the basis of the improvement in maximum F_n-D of each inlet over that of the rounded-lip inlet and expressed as fraction of ideal thrust. The values of pressure recovery and drag coefficient shown in the figure are the values at the diffuser discharge Mach number corresponding to peak thrust minus drag. At the extreme right of the figure are shown data for the normal-wedge inlet which has not been previously discussed in this paper. This inlet features a compression wedge normal to the surface of the fuselage and at this Mach number had an included angle of 28°. The semicircular cowl was swept behind the shock angle from the wedge at a free-stream Mach number of 2.0 and had a relatively sharp profile. In order to permit accomodation of this inlet to the test configuration, it was necessary to reduce the inlet size to 80 percent of the design value. In addition, data for the normal wedge inlet were not obtained at values of $\frac{h}{\delta} = 1.0$. Thus the actual data shown by the solid lines were extrapolated and are shown as dashed lines to represent an inlet size and bleed scoop height comparable to the other inlets.

The increase in thrust minus drag obtained by sharpening the lip of the semicircular cowl (equal to 7 percent of the ideal thrust) represents a gain of approximately 1300 pounds in available thrust for the entire airplane. Of the sharp-lip inlets, the most efficient is the rectangular cowl with side fairings which indicates an improvement of 13 percent of the ideal thrust over the rounded-lip-inlet performance. Since, during the investigation, no attempt was made to optimize the performance of each of the inlets, it is felt that, with greater attention to design details, the performance of all of the sharp-lip inlets could be raised to a comparable level. Of perhaps greatest significance from the figure is the pronounced superiority of all sharp-lip inlets over the rounded-lip inlet at a Mach number of 2.0.

This marked superiority disappears, however, at Mach number 1.5 as is shown by the comparison in figure 7 of the on-design performance of the various inlets at that Mach number and at 3° angle of attack. Here the maximum variation in performance of all inlets amounts to only 3 percent of the ideal thrust. Thus, use of rounded-lip inlets for the flight range to $M_0 = 1.5$ appears satisfactory. This is confirmed somewhat by the data of reference 6 which indicate that lip profile is not critical at low supersonic Mach numbers as regards pressure recovery.

For a complete evaluation of the relative merits of the various inlets, consideration must be given not only to the aerodynamic performance at the supersonic Mach numbers but there must also be a concern for the effects of angle of attack and operation at subsonic Mach numbers.

The sensitivity of the various inlets to angle of attack is illustrated in figure 8 for the design Mach number of 2.0. The ratio of the total pressure P_2 at any angle of attack to P_2 at 3° angle of attack is plotted as a function of the angle of attack. Normal wedge data are not shown because of the impracticability of extrapolating the performance at angle of attack to an h/δ of 1.0. The reference curve of the figure empirically predicts the decrease in the pressure recovery at angle of attack by considering only the isolated effect of the increasing boundary layer and consequent decrease in h/δ at angle of attack.

The radical departure of the recovery ratio of the rectangular cowl inlet with side fairings and of the conical spike inlet from this reference curve indicates the greater sensitivity of these inlets to cross-flow effects. (To a small degree, the conical spike inlet is penalized at angles of attack since it was alined with the fuselage axis.) Apparently, greater attention should be given to the circumferential location of these inlets as compared to that needed for the relatively insensitive rounded-lip inlet.

Considering pressure recovery P_2/P_0, relatively good performance is shown at 12° for the rounded-lip inlet; however, at angle of attack, the comparison of inlet performance should also include the effects of drag. This is done in figure 9 where the thrust minus drag is expressed as a fraction of the ideal thrust at $M_0 = 2.0$. The drag data at angle of 2509
attack lose considerable significance since the configuration does not represent the entire airplane. Therefore the ordinates of this figure are omitted and only relative comparisons can be made. Assessing the inlet merits in this manner shows the consistent superiority of the sharp-lip inlets at angle of attack to 12°. The relatively greater sensitivity to angle of attack of the rectangular cowl inlet with side fairings and of the conical spike inlet is again emphasized.

Pressure-recovery characteristics of several of the inlets were obtained at a Mach number of 0.64. These data are shown in figure 10 where the pressure recovery is plotted as a function of the diffuser discharge Mach number. The data were obtained with the inlets designed for $M_0 = 1.5$ and subsequently extrapolated to represent the condition of minimum displacement position of the compression surfaces. Drag data were not obtained at this Mach number. Thus, comparison of inlet performance can best be made by reference to the pressure recoveries at the diffuser discharge Mach number that would match the previously assumed engine, operating at an altitude of 35,000 feet. On this basis, the differences in pressure recovery of the inlets investigated were relatively small. Although not tested at this Mach number, the inlet having a rounded lip and semicircular cowl would be expected to give comparable or better pressure recoveries than any of the inlets shown in this figure. It therefore appears that, from the standpoint of pressure recovery at this Mach number as at low-supersonic Mach numbers, there is little to choose between sharp- and rounded-lip profiles. The data of reference 6 also show that sharp profiles and rounded profiles can give comparable performance for high-subsonic Mach numbers.

Take-off performance of the various inlets was simulated by a bench test of the configurations. These data are shown in figure 11 in the form of total pressure recovery plotted against diffuser discharge Mach number. As at the subsonic Mach number, the compression surfaces were here assumed to be at the minimum displacement giving effectively maximum inlet area. As might well be expected, the sharp-lip inlets suffer considerable pressure-recovery losses compared to the rounded-lip inlet. Shown in the figure is the diffuser discharge Mach number

required for engine matching at sea level for static conditions. The pressure recoveries of the various inlets at the engine-matching Mach number are translated to the ratio of thrust divided by engine rated thrust and shown to the right of the designations of the various curves. A high percentage of this maximum available thrust can be realized by the rounded-lip inlet, showing a value of 92 percent. In contrast, the best sharp-lip design, that of the semicircular cowl, obtains only 74 percent. For sharp-lip inlets designed primarily for high supersonic
2509 performance, the use of auxiliary air intakes thus appears highly necessary unless large losses in maximum take-off power can be tolerated.

In conclusion, the results of this investigation indicate: (1) Rounded-lip inlets may give satisfactory performance up to a Mach number of 1.5. (2) The use of sharp-lip inlets can result in considerable gain in available thrust at $M_0 = 2.0$ but has the added complication of requiring auxiliary air intakes for take-off conditions. (3) No inlet type is markedly superior over the entire range of operating variables so that the choice of a specific inlet design will be influenced by consideration of structural and mechanical complexities as well as the aerodynamic performance.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio

REFERENCES

1. Wittliff, Charles E., and Byrne, Robert W.: Preliminary Investigation of a Supersonic Scoop Inlet Derived from a Conical-Spike Nose Inlet. NACA RM L51G11, 1951.

2. Goelzer, H. Fred, and Cortwright, Edgar M., Jr.: Investigation at Mach Number 1.88 of Half of a Conical-Spike Diffuser Mounted as a Side Inlet with Boundary-Layer Control. NACA RM E51G06, 1951.

3. Luskin, H., and Klein, H.: High Speed Aerodynamic Problems of Turbojet Installations. Rep. No. SM-13830, Douglas Aircraft Co. (Santa Monica), Sept. 1, 1950.

4. Schueller, Carl F., and Esenwein, Fred T.: Analytical and Experimental Investigation of Inlet-Engine Matching Problems for Turbojet-Powered Aircraft at Mach Numbers up to 2.0. NACA RM E51K20, 1952.

5. Blackaby, James R.: An Analytical Investigation of Air-Induction Systems for Turbojet-Powered Airplanes Designed to Operate at Mach Numbers up to 1.50. NACA TN (to be pub.)

6. Hayes, Clyde: An Investigation of Circular Nose Inlets and Half-Circular Scoop-Type Inlets Designed to Meet Typical Turbojet Requirements through the Mach Number Range from 0 to 2. NACA RM (to be pub.)

7. Watson, Earl C.: Some Low-Speed Characteristics of an Air-Induction System Having Scoop-Type Inlets with Provisions for Boundary-Layer Control. NACA RM A51F15, 1951.

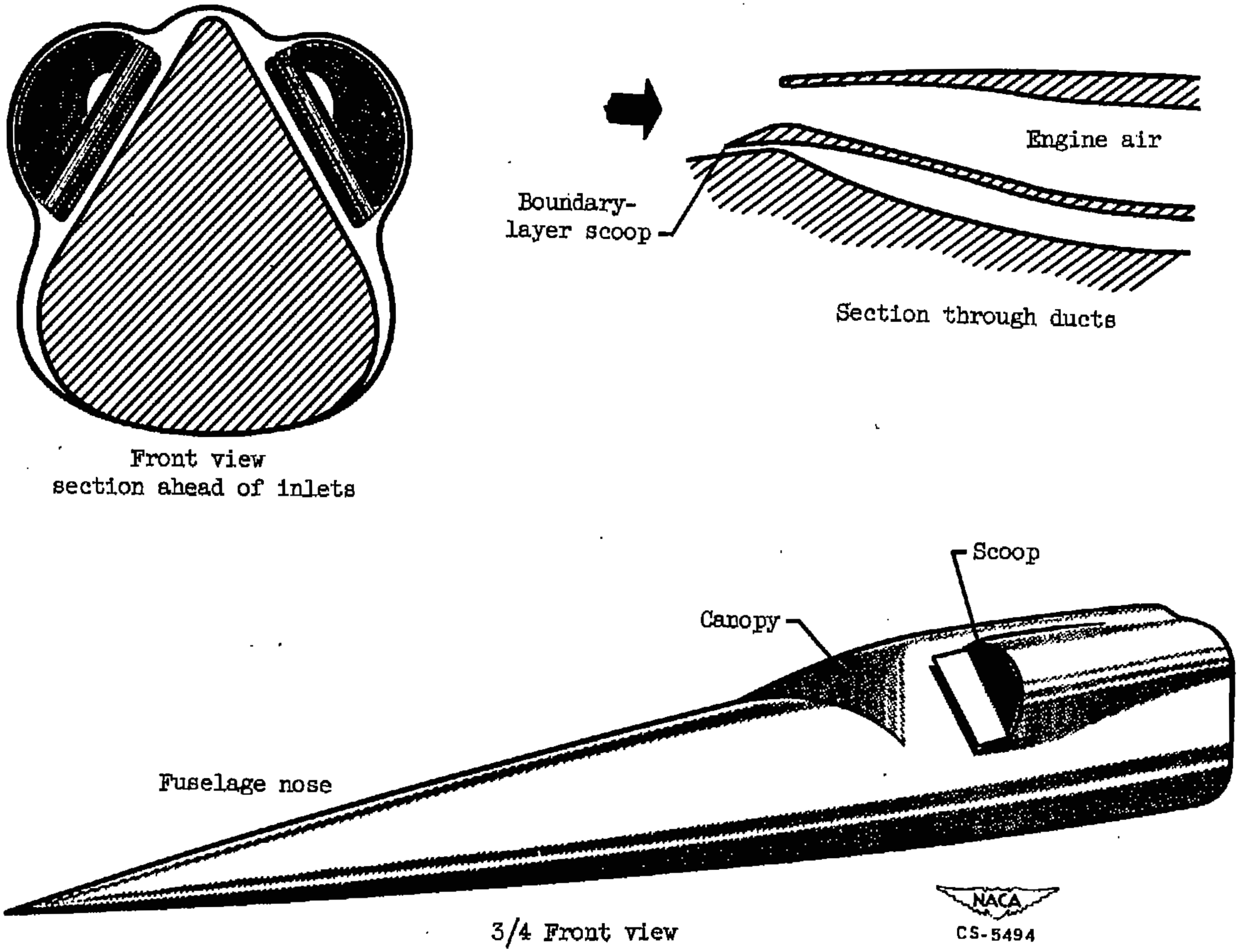

Figure 1. - Model configuration.

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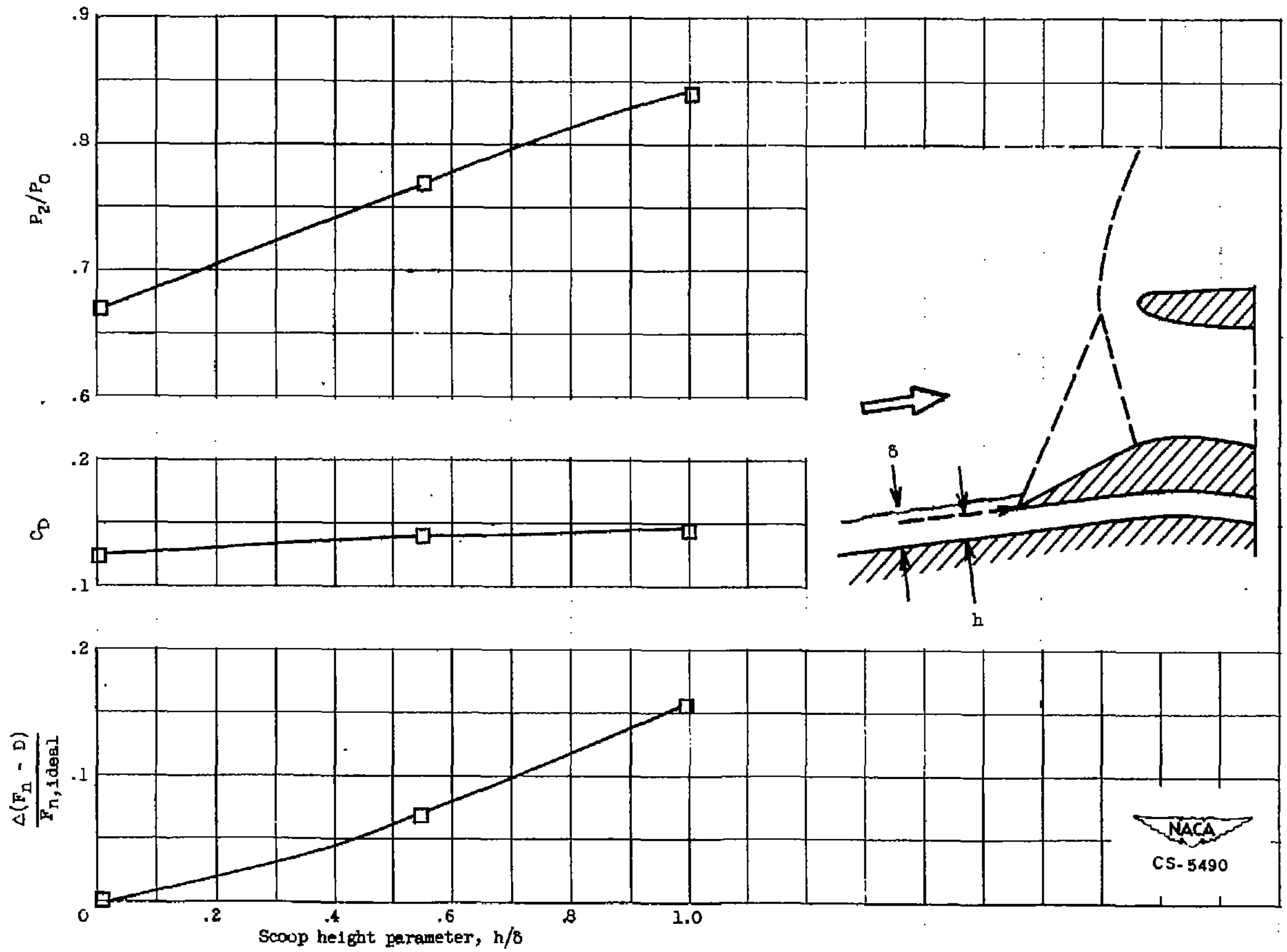

Figure 2. - Effect of boundary layer removal on inlet performance. M_0, 2.0; α, 3°.

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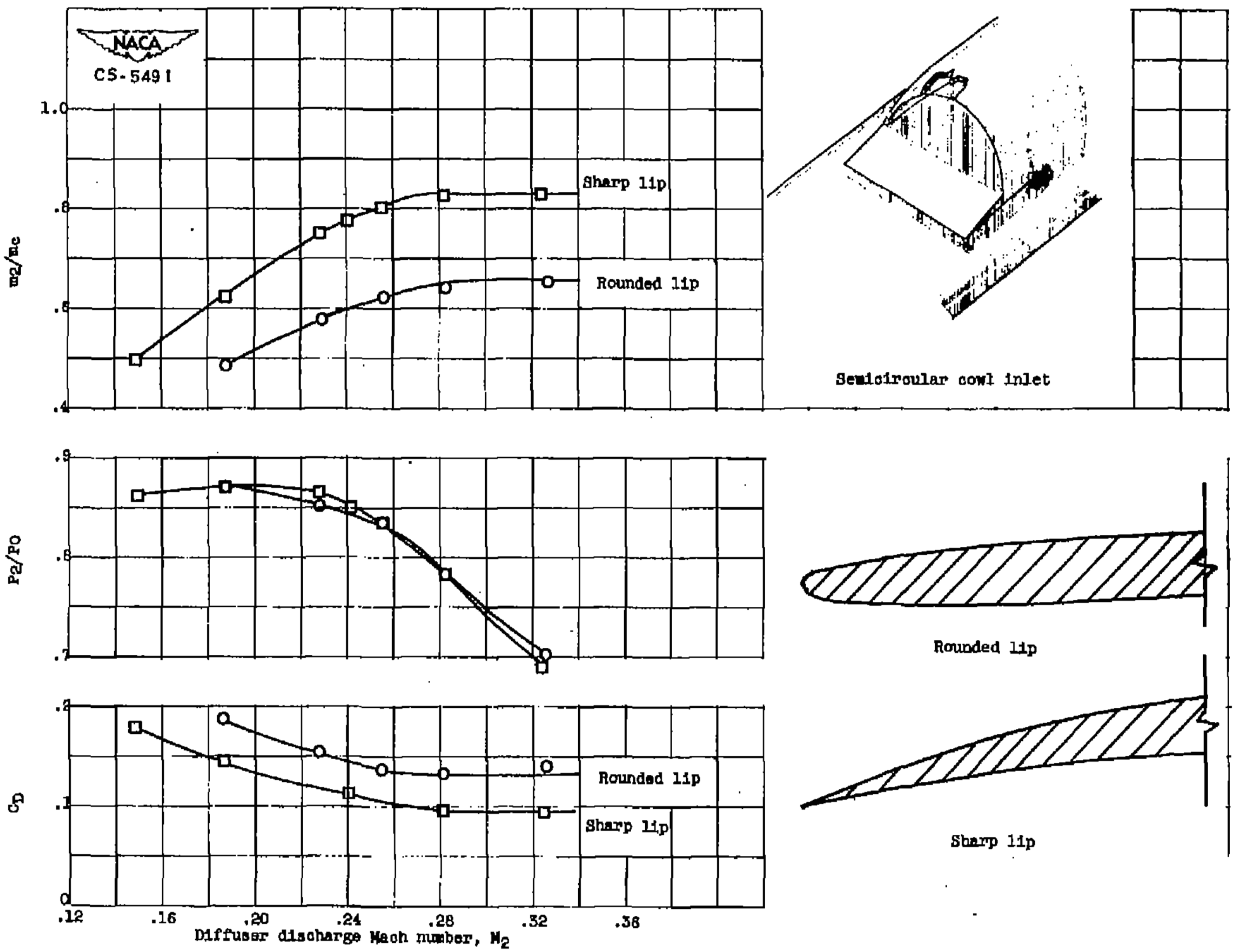

Figure 3. - Effect of lip profile on inlet performance. M_0, 2.0; α, 3°.

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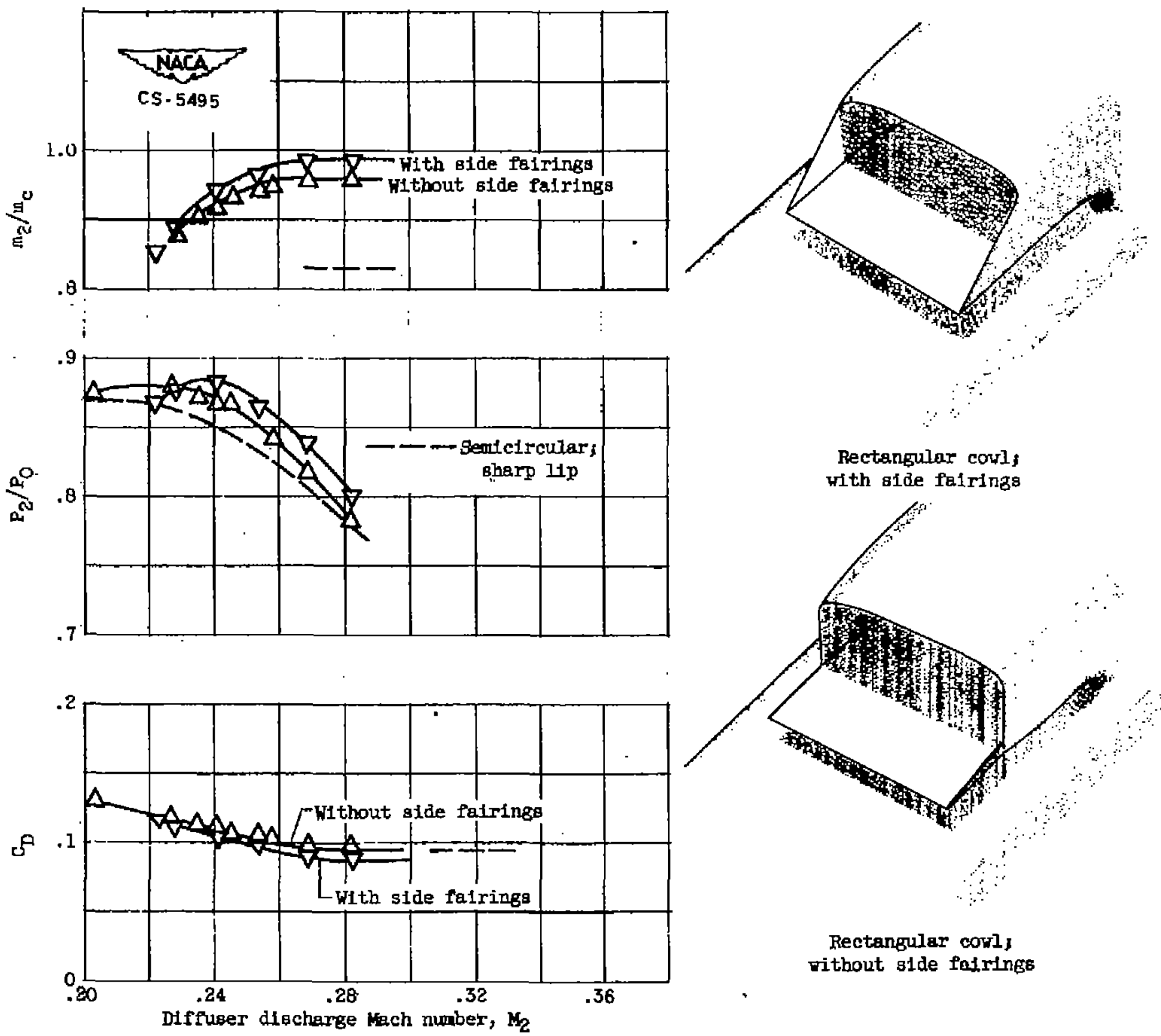

Figure 4. - Effect of cowl shape on inlet performance. M_0, 2.0; α, 3°.

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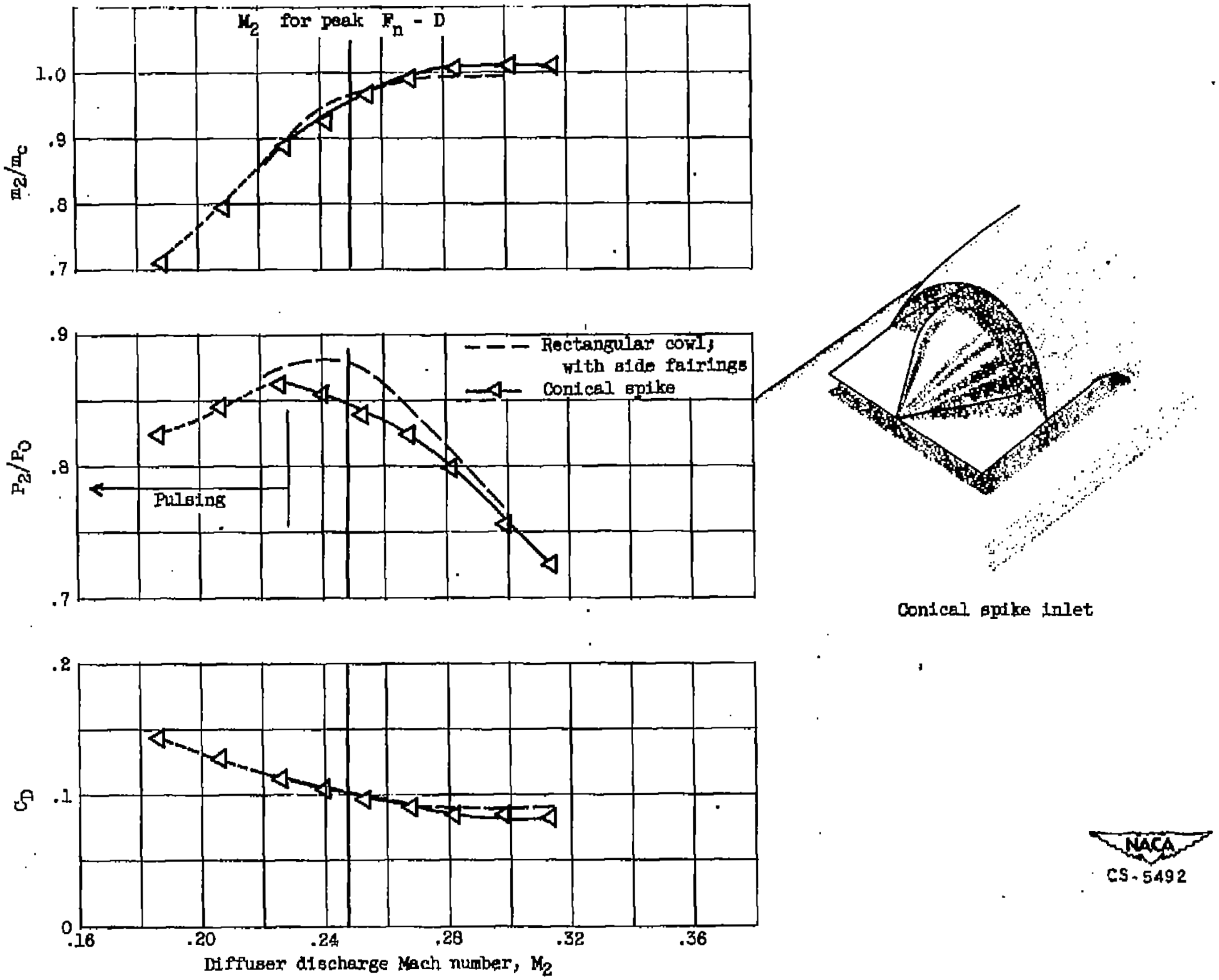

Figure 5. - Performance characteristics of inlets designed for minimum spillage. M_0, 2.0; α, 3°.

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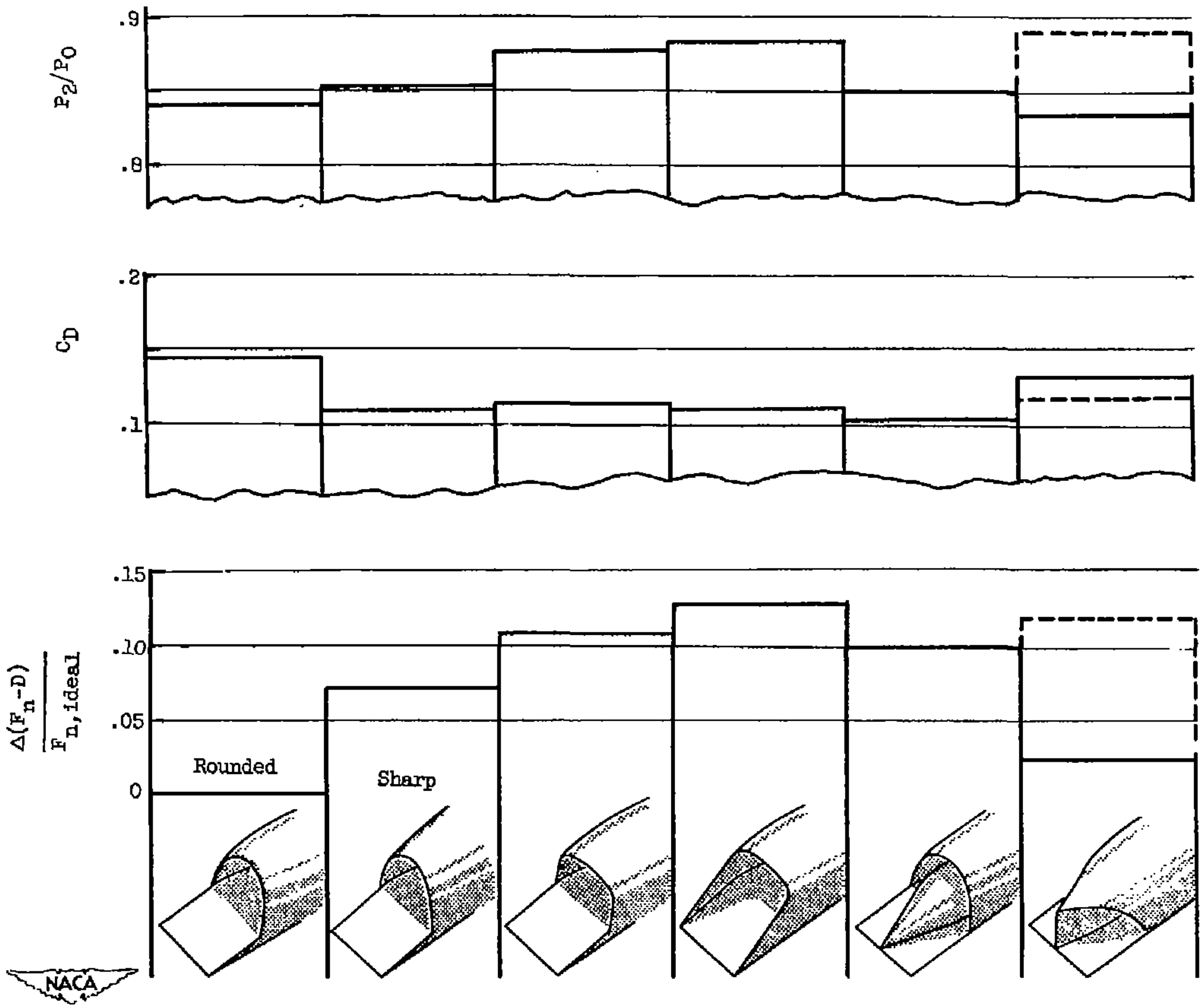

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Figure 6. - Comparison of inlet performance at maximum thrust-minus-drag. M_0, 2.0; α, 3^o.

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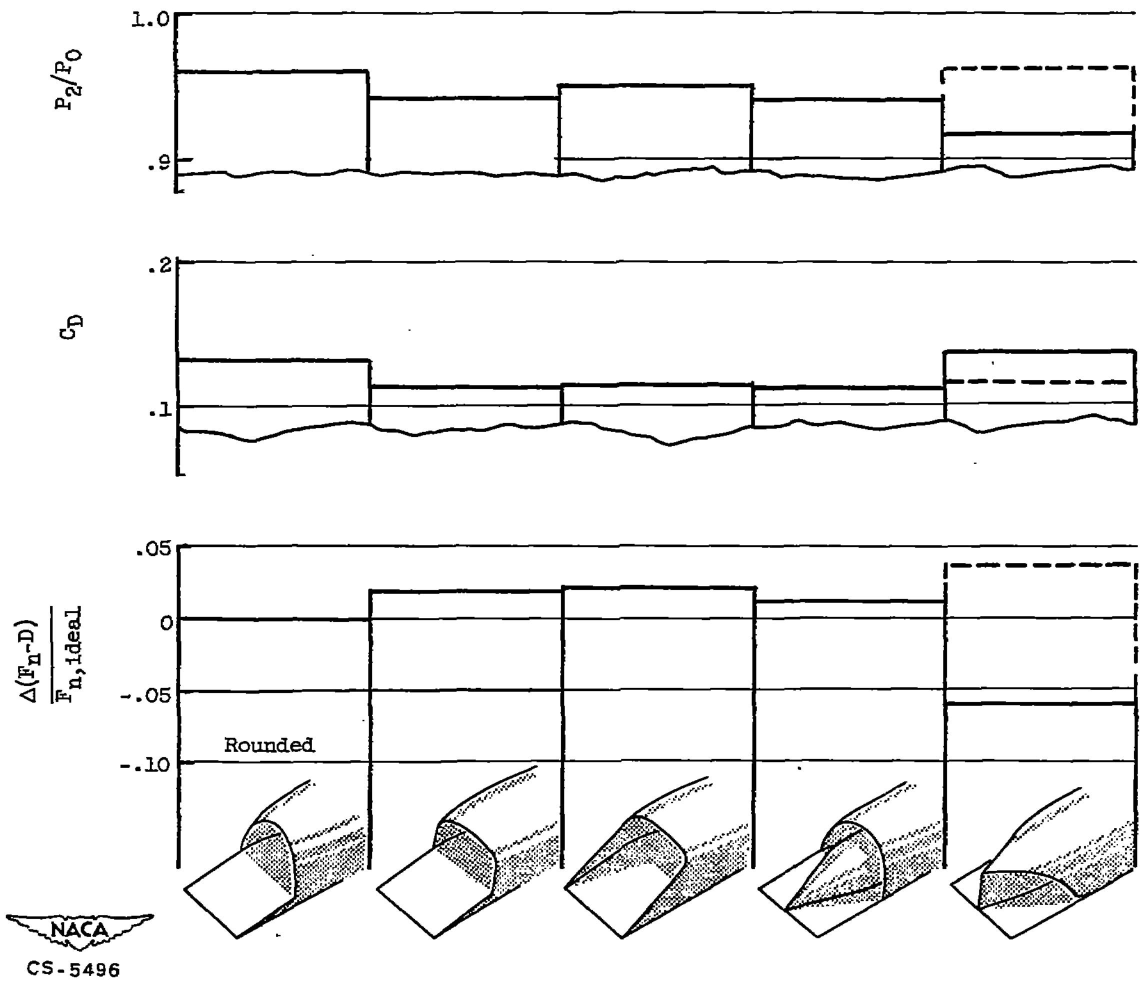

Figure 7. - Comparison of inlet performance at maximum thrust-minus-drag. M_0, 1.5; α, 3°.

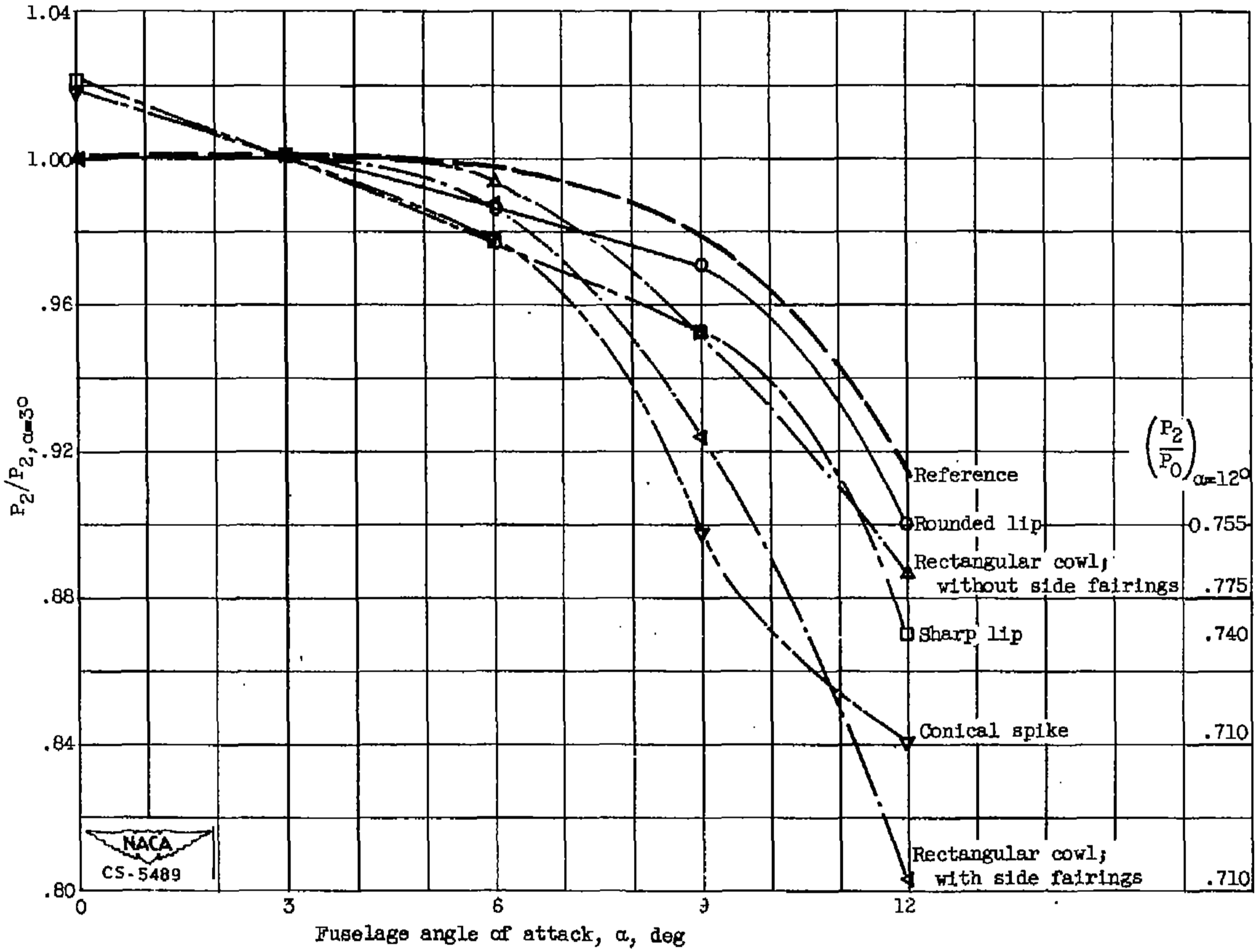

Figure 8. - Effect of angle of attack on total pressure recovery. M_0, 2.0; α, 3°.

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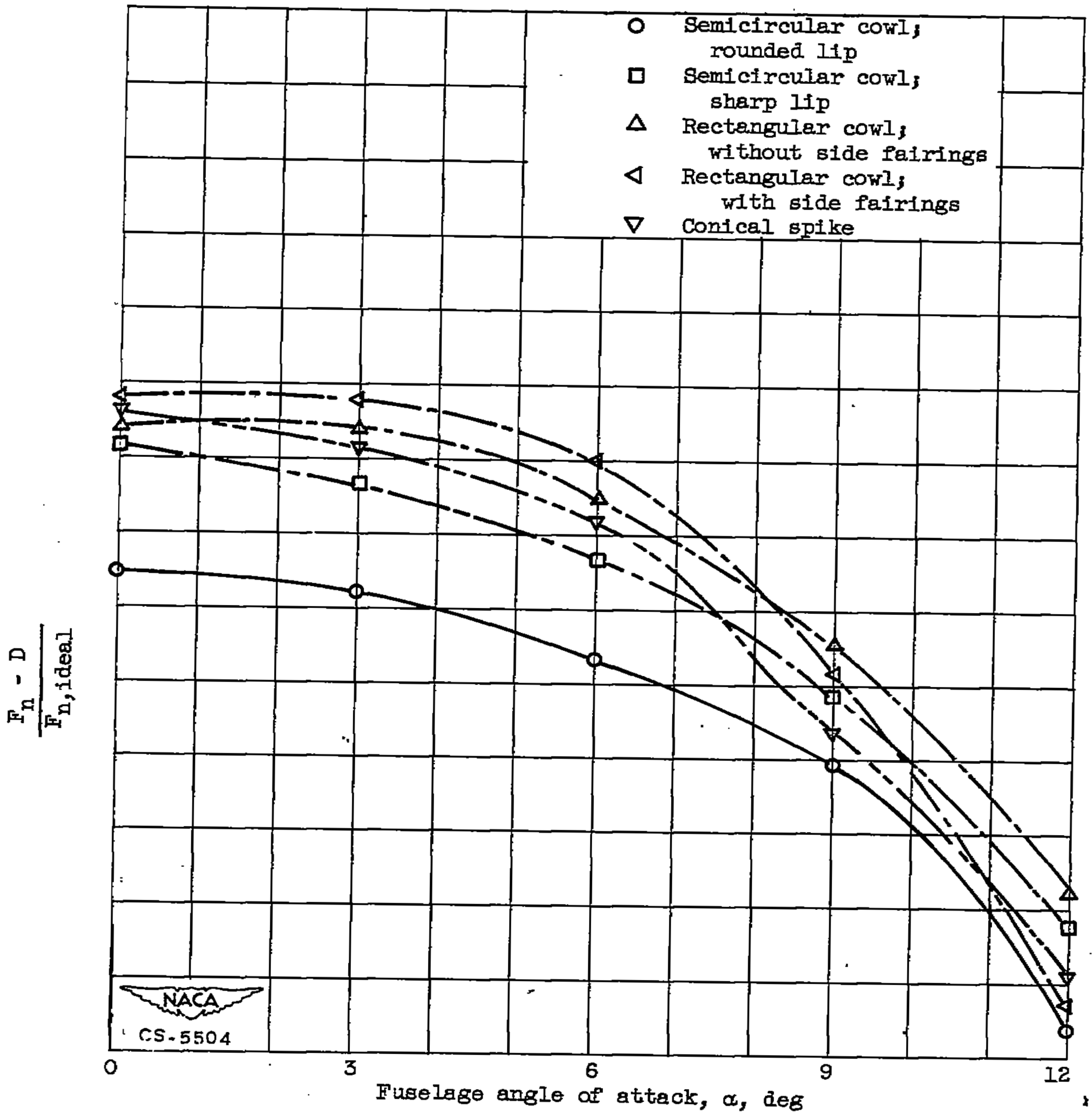

Figure 9. - Effect of angle of attack on thrust-minus-drag. M_0, 2.0; α, 3°.

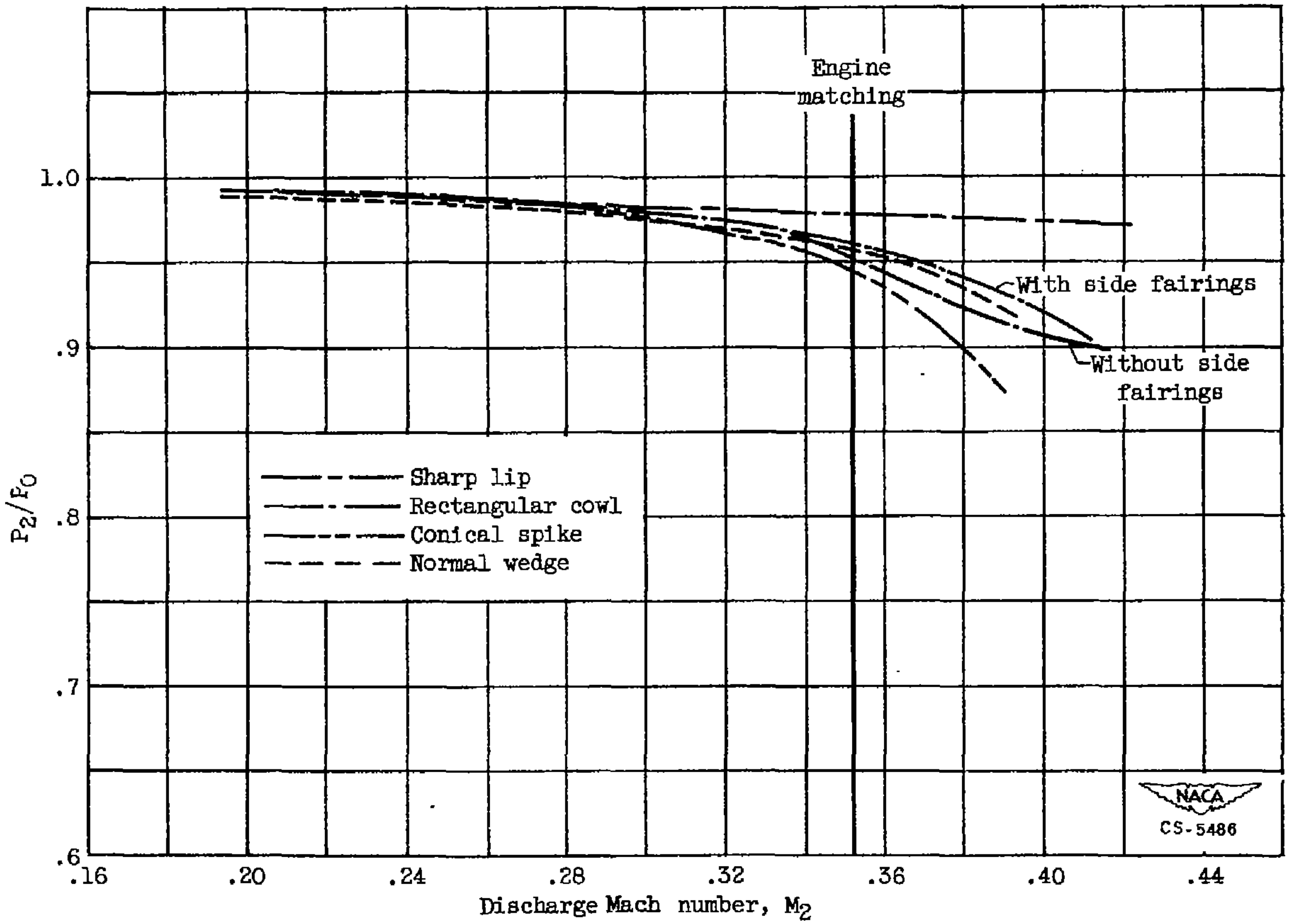

Figure 10. - Variation of total pressure recovery with diffuser discharge Mach number at subsonic free-stream Mach number, M_0, 0.64; α, 3°.

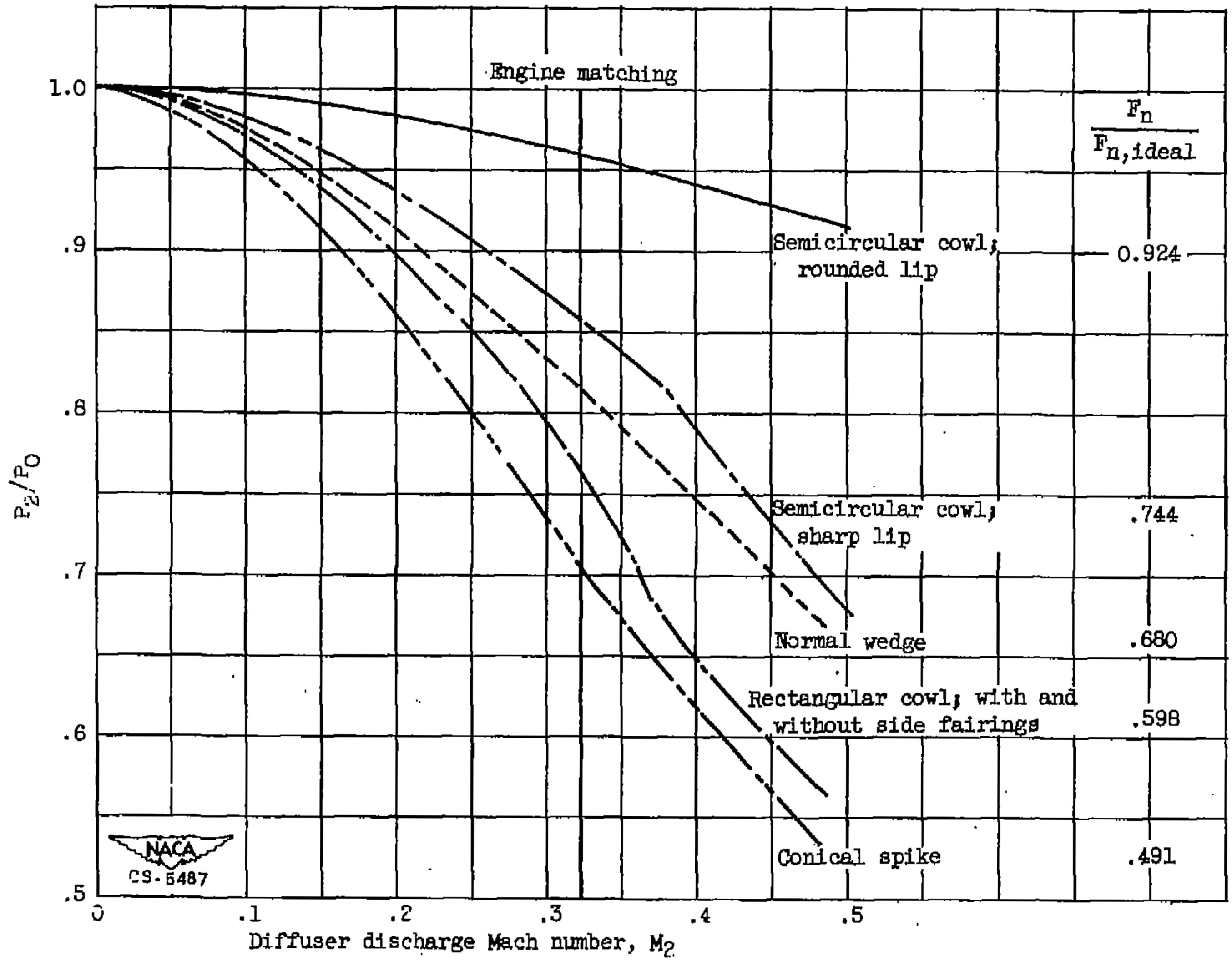

Figure 11. - Variation of total pressure recovery with diffuser discharge Mach number for take-off condition.

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